(NASA-CR-163197) AN ANALYTICAL STUDY OF EFFECTS ON AEROELASTICITY ON CONTROL EFFECTIVENESS Final Report, Jun. 1974 - Mar. 1980 (Kansas Univ. Center for Research, Inc.) 14 p HC A02/MF A01 CSCL 01C G3/08

N80-24322

Unclas 20981

THE UNIVERSITY OF KANSAS CENTER FOR RESEARCH, INC.
2291 Irving Hill Drive—Campus West
Lawrence, Kansas 66045

AN ANALYTICAL STUDY OF EFFECTS OF AEROELASTICITY ON CONTROL EFFECTIVENESS

Final Report

Prepared by

Sudhir C. Mehrotra
Flight Research Laboratory
University of Kansas Center for Research, Inc.
Lawrence, Kansas 66045

March 1980

Prepared Under NASA Grant NSG-1046

THE UNIVERSITY OF KANSAS CENTER FOR RESEARCH, INC.

2291 Irving Hill Drive— Campus West
Lawrence, Kansas 66045

Introduction

The purpose of this report is to inform NASA Headquarters and NASA-Langley on the work completed on grant NSG-1046 entitled, "An Analytical Study of Effects of Aeroelasticity on Control Effectiveness", during the period June 1974 through March 1980. This is the final report submitted to NASA on this grant.

Completed Work and Conclusions

The description and conclusion of the completed work is divided into ten small studies.

1. Effect of Changing Constant Percent Chordwise Lines on ΔC_p Values

This study was conducted to find the best paneling scheme which could be used in the Elastic Stability Derivative (ELASTAD) program of Reference 1. This program predicts ΔC_p values over two and three dimensional elastic wings, with and without camber and flap, at subsonic and supersonic speeds. The program of Reference 1, based on the theory of Reference 2, uses equidistant or almost equidistant constant percent streamwise lines (CPSWL's). Therefore, this study considers only the effect of changing constant percent chordwise lines (CPCWL's) on ΔC_p values.

Several different chordwise paneling schemes were used, on two and three dimensional wings with and without flaps, to predict ΔC_p distributions using ELASTAD program. These ΔC_p distributions were compared against analytical and/or experimental distributions. The best ΔC_p distributions were obtained by using the Modified Woodward Scheme for constant percent chordwise lines with the control point

located at "0.85" of the local panel chord. In the Modified Woodward Scheme the panels nearest the leading and trailing edges are half the size of the panels in-between. For wings with flaps, the Modified Woodward Scheme should be used separately, ahead and behind the flap hinge line. ΔC_p distributions over thick airfoils, like the GA(W)-1, should be used with caution because at non-zero angles of attack the predicted ΔC_p values are higher at the trailing edge than those obtained by experiment. At zero angle of attack, the difference between predicted and experimental ΔC_p values is significant both at the leading and the trailing edge.

The details of this study are given in reference 3. Further work on control point location is reported in section 9.

2. Structural Complexity Study for Evaluation of Structural Influence Coefficient Matrices

Several wings of either solid or built-up type construction were used for this study, the details of which are given in reference 4. The structural influence coefficient matrices for these wings were evaluated by using ELASTAD- and NASTRAN-program (ref. 5). ELASTAD program uses slender beam theory to represent the wing, whereas NASTRAN uses the actual elastic properties of the wing.

In ELASTAD program the wing is represented by an elastic axis, thus the input data needed for a wing is,

a. Unit loading point locations.

b. Elastic axis coordinates.

c. Bending stiffness (EI) of elastic axis segments.

d. Torsional stiffness (GJ) of elastic axis segments.

e. IASIGN-array, which assigns aerodynamics panels (loading points) to elastic axis end points.

A computer program was written to provide elastic axis coordinates and EI- and GJ-values for its segments for solid wings (see Appendix in ref. 4). The unit loading point locations and IASIGN-array was evaluated by hand for solid wings. For built-up wings, all the input data was hand calculated.

In NASTRAN program, the solid wings were assumed to be composed of several triangular and/or quadrilateral plate elements. An average thickness for each element was calculated by dividing its volume by its planform area. The built-up wings were modeled by using triangular and/or quadrilateral plates, shear panels and rods. The upper and lower surfaces were represented by plates, the spars and ribs by shear panels, and the flanges by rods.

Most of the experimental results presented deflection influence coefficients (DIC's) only. The ELASTAD program, which uses slender beam theory, calculates streamwise rotational influence coefficients (RIC's); whereas, the NASTRAN program, which employs the actual elastic properties of the wing, calculates both DIC's and RIC's. The appropriate comparisons for all the wings are shown in reference 4.

In case of solid wings, the DIC's obtained by using NASTRAN program matched reasonably well with experimental measurements. In cases of built-up wings, the NASTRAN predicted slightly lower deflections than the experiment. These larger experimental deflections could be due to the mechanical construction of the wings. Rotational influence coefficients obtained by using ELASTAD and NASTRAN programs do not match exactly at any particular point on all the wings studied. The smallest difference between the two sets of rotations occurs, for untapered solid- and built-up - wings of aspect ratio between 2 and 6, when load- and rotation-points are on the elastic axis. For the load or rotation-points away from the elastic axis, this difference increases but is still reasonable. The larger difference is due to the SBM assuming the rigid links between the elastic axis endpoints and the load- and rotation-points; whereas, the NASTRAN employs the actual elastic properties of the structure. Thus, it is suggested to use the NASTRAN program for evaluating the RIC's to be used in ELASTAD program.

3. Arrow Wing Study

H. W. Carlson concluded from a study of arrow wings (ref. 6) that the discrepancy between the experimental and linearized theory estimates of pressure distribution might be due to aerolastic deflections and the presence of vortex flow. An arrow wing of symmetric airfoil section of reference 6 was studied for its aeroelastic effects (see ref. 7).

The structural influence coefficient matrix for this wing was calculated by using the NASTRAN program (ref. 5). This was done by dividing the wing into triangular- and quadrilateral-plate elements.

The NASTRAN program can only handle constant thickness plate elements and so an average thickness of each element is used for calculating the structural influence coefficient matrix. This matrix was manipulated to generate another matrix which conformed to the aerodynamic paneling. Mass of each of the panel was calculated by multiplying its area to mean thickness and density.

All the above mentioned information was used in ELASTAD program to calculate rigid and elastic ΔC_p distributions at the experimental angle of attack (8 degrees) and Mach number (2.05). The difference between the rigid and elastic ΔC_p's was small, which meant elastic effects were small. These ΔC_p's were quite different from those measured experimentally.

It was of interest to compare the streamwise rotations of panels obtained theoretically, by using ELASTAD program, and those obtained by employing experimental ΔC_p loading. The difference between the two sets of rotations is small at the root and large at the tip. The largest rotation of 1.94 degrees was predicted by experimental loading and 3.06 degrees by ELASTAD program. Both experimental and theoretical rotations are small which means elastic effects are small. Thus, the discrepancy between the experimental and theoretical ΔC_p values can be attributed to the vortex flow.

4. Modification of ELASTAD Program

Geometry part (AEREAD) of ELASTAD program has been modified to plot top and side views and structural part (AERELAS) to plot panels and elastic axis of the models. Until recently IASIGN - array had to be done graphically and punched for use in ELASTAD. A subroutine has been written and tested

to do this automatically. This makes input data preparation easier and panel assignment consistent.

5. Structural Matrix Conversion Package

A computer program was written to convert a structural influence coefficient matrix from its structural network to the one which conforms to the aerodynamic paneling of ELASTAD program. The details of this program are given in reference 8. This program was used in Transonic Aircraft Technology project, which is described next.

6. Participation in the Transonic Aircraft Technology (TACT) Project

In this project the rigid and elastic stability derivatives were caldulated for TACT aircraft for the following flight conditions:

$$M = 0.60,\ q_\infty = 300, 500, 533, 600, 825, 1050 \text{ lbs/ft}^2$$

$$M = 0.85,\ q_\infty = 300, 600, 825, 1050 \text{ lbs/ft}^2$$

$$M = 0.90,\ q_\infty = 300, 600, 825, 1050 \text{ lbs/ft}^2$$

All the stability derivatives are listed and plotted in reference 9. The following observations are noted from the results.

The experimental rigid lift curve slope (C_{L_α}) is always higher than that calculated by ELASTAD and the difference between the two increases with Mach number. At $M = 0.6$, the difference between the experimental and ELASTAD values is 5 percent which is assumed to be in the error bound, but at $M = 0.85$, the difference increases to 10 percent. The difference between experimental and ELASTAD numbers could be due to unmodeled transonic effects being significant and possibly the vortex flow due to strake type behavior of the planform. At a constant Mach number elastic C_{L_α} and lift coefficient for zero angle of attack (C_{L_0})

decrease with dynamic pressure. This is due to elastic unloading of the aircraft for swept-back wings and results from the fact that the increase in local α associated with twisting is overcome by the decrease in local α due to bending. The non-zero mass C_{L_α} is higher than zero mass C_{L_α}. These higher magnitudes of C_{L_α} indicates that the flap region of the wing and tail have larger mass concentration and so the increase in local α due to twist is higher than decrease in local α due to bending as compared to zero mass C_{L_α}. This could also be explained by examining the variation of lift coefficient with load factor $\frac{dC_L}{dn}$ which are positive. A positive value of $\frac{dC_L}{dn}$ indicated that the lift coefficient increases as the load factor increases, which, for this wing, could happen only when the flap region and tail are heavier. When a positive load factor is applied to such an aircraft, the local α increases due to the inertial forces which act opposite to the direction of motion. Thus, the lift is also increased. The equation for non-zero mass C_{L_α} (Ref. 9) suggests that for a positive value of $\frac{dC_L}{dn}$, the non-zero mass C_{L_α} is always higher than zero mass C_{L_α}.

The experimental rigid pitching moment curve slope (C_{m_α}) is always less negative than the one calculated by ELASTAD and the difference between the two increases with Mach number. At M = 0.6, the difference between the experimental and ELASTAD values is 3 percent, but at M = 0.85 the difference is 60 percent. The difference between experimental and ELASTAD numbers could be due to the same unmodeled effects previously mentioned. At a constant Mach number, the zero-mass elastic C_{m_α} becomes

less negative as the dynamic pressure increases and the pitching moment coefficient for zero angle of attack (C_{m_0}) is less negative than the rigid value at low dynamic pressure, but becomes more negative at higher dynamic pressures. These variations are again due to elastic unloading of the aircraft with swept-back wings. The non-zero mass C_{m_α} is always more negative than zero mass C_{m_α}. The same logic of flap region and tail having larger concentration of mass applies here also. The variable to be noted here is the variation of pitching moment coefficient with load factor $\frac{dC_m}{dn}$, which is negative. A negative value for $\frac{dC_m}{dn}$ always means that non-zero mass C_{m_α} will be more negative than zero mass C_{m_α}, unless $\frac{dC_L}{dn}$ is larger than the lift coefficient for trimming which is highly unlikely because $\frac{dC_L}{dn}$ is always a small number.

The experimental rigid static margin ($C_{m_\alpha}/C_{L_\alpha}$) is always less negative stable than the one calculated by ELASTAD and the difference between the two increases with Mach number. At $M = 0.6$, the difference between the experimental and ELASTAD values is 10 percent, but at $M = 0.85$ the difference is 67 percent. The difference between experimental and ELASTAD numbers could be due to the same unmodeled effects previously mentioned. At a constant Mach number, the zero-mass elastic static margin becomes less negative as the dynamic pressure increases. The non-zero mass static margin is always more negative than zero mass static margin indicating that the effect of masses is to stabilize the aircraft.

7. Extension of Structural Influence Coefficient Matrix Evaluation Package to Swept-forward Wings

Two errors in the ELASTAD program were detected during a routine calculation of a structural influence coefficient matrix. These were:

a. Wrong coding of few Fortran statements

b. Wrong definition of one of the input variables

After making these modifications, it was realized that ELASTAD should be modified further to include the evaluation of structural influence coefficient matrix for swept forward wings and for the cases when the panel controids on the horizontal tail lie behind the last panel centroid of the fuselage. All these modifications have been checked out.

8. Leading-Edge Vortex Separation Study

A numberical method is developed to predict distributed and total aerodynamic characteristics for low aspect-ratio wings with partial leading-edge separation. The flow is assumed to be steady and inviscid. The wing boundary condition is formulated by the Quasi-Vortex-Lattice method. The leading-edge separated vortices are represented by discrete free vortex elements which are aligned with the local velocity vector at mid-points to satisfy the force free condition. The wake behind the trailing-edge is also force free. The flow tangency boundary condition is satisfied on the wing, including the leading-and trailing-edges. Comparison of the predicted results with complete leading-edge separation has shown reasonably good agreement. For cases with partial leading-edge separation the lift is found to be highly nonlinear with angle of attack.

The theoretical details of this study are given in reference 10 and the computer program in reference 11. This program was recently described in the Fall 1979 issue of the quarterly publication "NASA Tech Briefs".

9. Evaluation of Wing-Tip-Suction at Subsonic and Supersonic Speeds

The aerodynamic method which has been used in ELASTAD program is generally known as Woodward's panel method. A simplified version of ELASTAD program was modified to check out the concept of evaluating the leading-edge and side-edge suction forces.

Woodward's panel method for subsonic and supersonic flow is improved by employing control points determined by exactly matching two-dimensional pressure at a finite number of points. The results show great improvement in the predicted pressure distribution of a flapped airfoil. With the paneling scheme of cosine law in both chordwise and spanwise directions, the method is shown to accurately predict leading-edge and side-edge suction forces of various configurations in subsonic and supersonic flow.

Based on the extensive comparison of present prediction with other theoretical results, it may be concluded that the present improved Woodward's panel method is generally accurate in predicting the leading-edge and side-edge suction forces and the centers of these forces in subsonic and supersonic flow. The good accuracy of the present method has also been demonstrated for cambered and flapped airfoils. Because of generality of the panel method, the present improved method can therefore be used not only to predict the vortex lift of complex planforms through the method of suction analogy, but also to calculate certain lateral-directional stability derivatives as well.

The details of this study are given in references 12 and 13.

10. Evaluation of Deflection Influence Coefficient Matrix for Solid Wing Model

Several solid wing models, for which deflection data was available, were initially used for this study. Later, on a solid wing of transport

type aircraft was used to measure deflection for several load conditions. These wings were modeled on Structural Performance Analysis and Redesign (SPAR) program (ref. 14) by using first plate elements and then solid elements (ref. 15 and 16). The solid element representation of the wings resulted in a better correlation of measured and calculated deflections. By using solid element representation the deflection influence coefficient matrices can be calculated with about 5% error.

References

1. Roskam, J.; Lan, C.; and Mehrotra, S. C.: A Computer Program for Calculating α- and q- Stability Derivatives and Induced Drag for Thin Elastic Aeroplanes at Subsonic and Supersonic Speeds. NASA CR-112229, October 1972.
2. Woodward, F. A.: Analysis and Design of Wing-Body Combinations at Subsonic and Supersonic Speeds. Journal of Aircraft, Vol. 5, No. 6, Nov.-Dec. 1968.
3. Mehrotra, S. C.: An Analytical Study of Effects of Aerolasticity on Control Effectiveness. First Progress Report on NASA Grant NSG-1046, February 1975.
4. Mehrotra, S. C.: An Analytical Study of Effects of Aeroelasticity on Control Effectiveness. Third Progress Report on NASA Grant NSG-1046, December 1975.
5. MacNeal, R. H. (Editor): The NASTRAN Theoretical Manual. NASA SP-221(01), December 1972.
6. Carlson, H. W.: Pressure Distributions at Mach Number 2.05 on a Series of Highly Swept Arrow Wings Employing Various Degrees of Twist and Camber. NASA TND-1264, May 1962.
7. Mehrotra, S. C.: An Analytical Study of Effects of Aeroelasticity on Control Effectiveness. Fourth Progress Report on NASA Grant NSG-1046, June 1976.
8. Mehrotra, S. C.: An Analytical Study of Effects of Aeroelasticity on Control Effectiveness. Fifth Progress Report on NASA Grant NSG-1046, December 1976.
9. Mehrotra, S. C.: An Analytical Study of Effects of Aeroelasticity on Control Effectiveness. Sixth Progress Report on NASA Grant NSG-1046, June 1977.

10. Mehrotra, S. C.; and Lan, C. E.: A Theoretical Investigation of the Aerodynamics of Low-Aspect-Ratio Wings With Partial Leading-Edge Separation. NASA CR-145304, January 1978

11. Mehrotra, S. C.; and Lan, C. E.: A Computer Program for Calculating Aerodynamic Characteristics of Low-Aspect-Ratio Wings With Partial Leading-Edge Separation. NASA CR-145362, May 1978

12. Lan, C. E.; and Mehrotra, S. C.: An Improved Woodward's Panel Method for Calculating Edge Suction Forces. Journal of Aircraft, Vol. 16, No. 9, September 1979.

13. Lan, C. E.; and Mehrotra, S. C.: An Improved Woodward's Panel Method for Calculating Leading-Edge and Side-Edge Suction Forces at Subsonic and Supersonic Speeds. NASA CR-3205, November 1979.

14. Whetstone, W. D.: SPAR Structural Analysis System - Reference Manual. Vol. 1, NASA CR-145098-1, February 1977.

15. Mehrotra, S. C.: An Analytical Study of Effects of Aeroelasticity on Control Effectiveness. Tenth Progress Report on NASA Grant NSG-1046, June 1979.

16. Mehrotra, S. C.: An Analytical Study of Effects of Aeroelasticity on Control Effectiveness. Eleventh Progress Report on NASA Grant NSG-1046, December 1979.